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A Case History of Technology Transfer

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Introduction

The visualization of new concepts often occurs long before the technology is available to implement the concepts, or even before the devices are needed. Tracing the history of a program from concept to implementation can provide insight into the technology development process—including an awareness of that which was foresight, that which was luck, that which was logical planning, and that which resulted from economic or political conditions existing over the course of the development history. Wars have provided the incentive for major technology breakthroughs within a compressed development time span. The major technology developments occurring during the "Man on the Moon" U.S. space program are outstanding examples of technology moving forward in a timely manner.

Tracing the history of space electric rockets from Dr. Robert Goddard's description of their advantages in 1906 through Dr. Ernst Stuhlinger's first detailed discussion of electric rocket system analysis in 1954 to Dr. Harold Kaufman's testing of the first bombardment ion thruster illustrates the initial-concept-to-proof-of-concept process so common to a new technology.

The further development of the bombardment ion thruster after demonstration of concept followed the typical course: the flight test to prove it would work in space and to demonstrate long life; next some ups and downs in a varying economic, political environment; then a planned technology readiness program to focus the technology; and finally the orderly transfer of the technology from the research-technology NASA Center to the user-development NASA Center and their industrial team.

This case history of technology transfer traces some of the early history of the bombardment ion thruster and goes on to illustrate the approach used to ensure maximum yield from the technology program and transfer of that technology to the user.

Technology Transfer—A Process

Technology has been defined as the practical utilization of science. Few undertakings in the modern world are as important as the development of technology, yet few subjects are so little understood. The U.S. space program has caused great technological advances. As a result, men have climbed the lunar craters, robot spaceships have photographed the planets in our solar system, the world has been educated via communications satellites, land resource satellites help prospect for oil, and damaged hearts are run by pacemakers.

To do this, no other technological endeavor has set—and met—standards as high as those required in the space program. The term "zero defects" is an invention of space technology in which machines must function perfectly in extreme environments. It is ironic that this same requirement for zero defects has created a paradox wherein the most technologically innovative organization in the world is, at the same time, the most conservative about adopting new technologies to perform its missions in space.

All new technologies carry inherently higher risk than those that have demonstrated reliability and so are regarded by a project manager with, at best, suspicion and, at worst, total distrust. Technology transfer then is a multistep process. First, the mission advantages must be demonstrated to the potential user. Operational feasibility, functional reliability, and reproducibility within acceptable tolerances must be demonstrated. User familiarization and "hands on" hardware experience will then lead to the confidence about the technology so necessary for its acceptance.

The degree of confidence that must be demonstrated varies inversely with the need for the technology. Enabling technologies, those without which the mission cannot be performed, are most readily accepted, although sometimes with great trepidation. Technologies that merely enhance the mission capabilities are most strongly resisted until the enhancement clearly outweighs the potential risks. Always the project attitude is that it worked before and is reliable so don't fool with it—and rightly so.

This report describes a technology, electric propulsion, that remained in the realm of research and technology until two events occurred. One, the technologists at the NASA Lewis Research Center brought the hardware to a point where it performed reliably with a known and acceptable risk, and two, NASA Headquarters and the user Centers recognized this technology as enabling for a large class of missions that NASA is interested in undertaking. The

Marshall Space Flight Center was chosen to develop the system and the process of technology transfer began.

Electric Propulsion for Spaceflight

Spaceflight has been a reality for only a score of years, yet in that time enormous technological strides have been made. Unmanned exploration of the inner and outer planets has begun. The Voyager, Explorer, and Pioneer series spacecraft are yielding a wealth of knowledge about the nature of the solar system. Satellites in geosynchronous orbits have become a practical reality, providing physical data about the Earth and worldwide services in communications and navigation.

To date, virtually all propulsion systems for planetary or Earth orbital applications have been chemical devices. As progress continues in space, however, the missions will become more extensive and difficult to accomplish with only chemical propulsion because of the limited propulsion system mass that can be put into orbit. This report describes the history and development for one type of propulsion system that offers promise of augmenting the present chemical capabilities—an ion-thruster propulsion system. The focused-technology program developed to bring forth this new propulsion system is described, as is the process used to transfer this technology to the users.

The mercury-bombardment ion thruster has a propellant specific impulse of 3000 seconds. This impulse is approximately six to eight times that of the best chemical rocket systems. As is shown later the mass of the required electric powerplant does not permit a corresponding six- to eightfold increase in the velocity increment imparted to the spacecraft, but the gains are substantial. Thus the ion-thruster propulsion system opens new vistas to the mission planner and enlarges the total spectrum of achievable missions.

The principal elements of the ion-thruster propulsion system are shown in figure 1. The Sun provides energy, which is converted into electric power by solar cells. The power is then conditioned to the current and voltage needed by the ion thruster. Propellant is ionized in the thruster and electrically exhausted to produce thrust. For many missions the power source can serve the dual roles of providing both thruster power and power for mission objectives subsequent to the thrusting period. The thruster will be of appropriate size or numbers to satisfy the thrust requirements for the particular propulsion task.

Propulsion system mass has been largely that of its propellant. Raising the specific impulse (ratio of

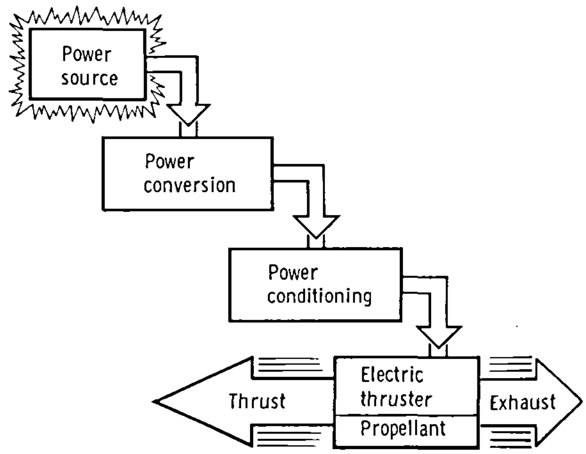


Figure 1. —Main subsystems of an electric propulsion system.

thrust to propellant mass flow) lowers the mass fraction of the system that is propellant. The main advantage of using electric propulsion is that the electric energy added to the exhaust propellant greatly increases its velocity, or specific impulse; hence the needed thrust is produced with a lower propellant flow rate. The mass of propellant required to produce a given total impulse decreases with increasing specific impulse, as shown in figure 2. The saving in propellant mass, however, is offset by the increasingly massive powerplant required to accelerate the exhaust to higher velocities. This increase in powerplant mass is shown in figure 3. The maximum payload of a spacecraft is achieved at the optimum specific impulse, where the sum of the propellant and powerplant masses is a minimum, as shown in figure 4.

As figure 4 indicates, at low specific impulse the propellant mass becomes excessive, and at high specific impulse the powerplant mass becomes excessive. Between these two extremes is a broad useful range where sufficient payload remains for design of a practical spacecraft. As defined by figure 4, total payload includes the mass of the spacecraft itself plus the mass of the useful payload. The optimum value of specific impulse to maximize payload usually is between 2000 and 5000 seconds,

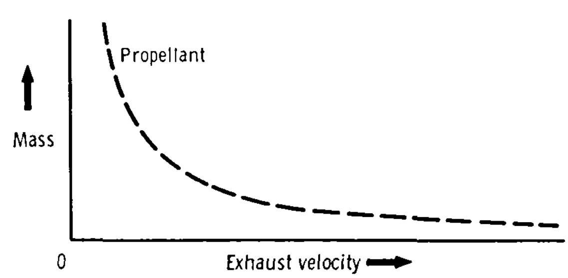


Figure 2. —Propellant mass as a function of exhaust velocity.

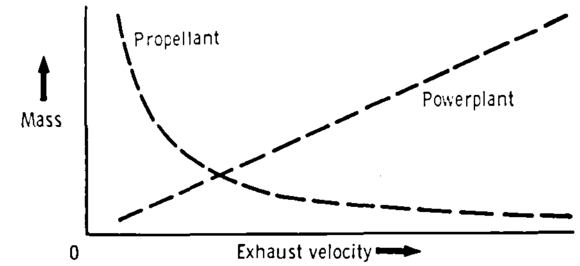


Figure 3.—Powerplant mass as a function of exhaust velocity.

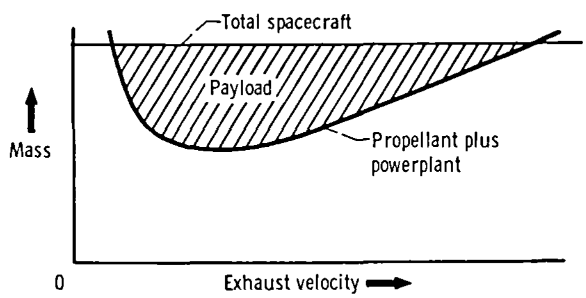


Figure 4.—Payload mass as a function of exhaust velocity.

and thus the optimum value of exhaust velocity is between 20 000 and 50 000 meters per second. This range of exhaust velocity is easily achieved with ion thrusters and, as is discussed later, results in large increases in spacecraft payload for a great variety of missions.

The entire process shown in figures 2 to 4 is influenced by the time over which the thrusters operate. Extending the operating time increases the total energy obtained from the solar cells and thereby increases the velocity increment imparted to the spacecraft. Because of this, electric propulsion systems operate continuously during the mission over long periods of time in order to extract the most energy from the lowest (therefore lightest) amount of solar cell power.

The first electron-bombardment ion thruster was conceived and tested by Dr. Harold R. Kaufman in 1959 at the NASA Lewis Research Center (ref. 1). This thruster operates by flowing a gaseous propellant into a discharge chamber. The propellant may be any gas, but mercury, cesium, and the noble gases are the most efficient for propulsion applications. Propellant atoms are ionized in the discharge chamber by electron bombardment in a process similar to that in a mercury arc sunlamp. This ionization occurs when an atom in the discharge loses an electron after bombardment by an energetic (40 eV) discharge electron. The electrons and the ions form a plasma in the ionization chamber. The electric field between the screen and the accelerator draws ions from the plasma. These ions are then accelerated out through many small holes in the screen and accelerator electrode to form an ion beam, as shown

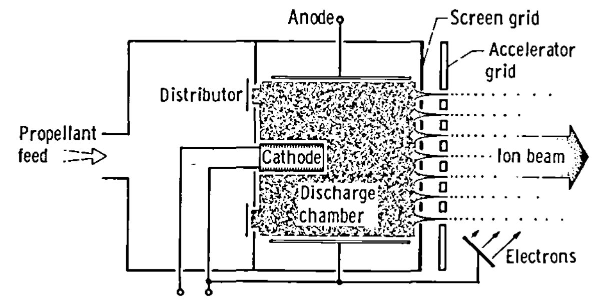


Figure 5.—Operation of electron-bombardment thruster.

in figure 5. A neutralizer injects an equal number of electrons into the ion beam. This beam of electrons allows the spacecraft to remain electrically neutral and is a requirement for successful thruster operation.

Technology development on the mercury-bombardment thruster has continued through the 1960's to the present time. Thrusters 2.5 to 150 centimeters in diameter have been successfully tested. These thrusters require power of 50 watts to 200 kilowatts and produce thrust of 0.4×10^{-3} to 4 newtons (0.1×10^{-3} to 1 lb).

The mercury-bombardment ion thruster technology developed at the NASA Lewis Research Center has been recognized throughout the world as establishing the state of the art for electric propulsion. Programs in England, Germany, and Japan all employ derivatives of the Lewis thruster design. Germany and Japan in particular have mounted aggressive programs leading to approved flight test programs in the early 1980's. In the United States, the Air Force has accepted the Lewis ion thruster as a potential solution for its future space propulsion requirements and is supporting a test flight in 1982 on the P80-1 satellite.

History of Electric Propulsion

Early Concepts

Dr. Robert H. Goddard in 1906 and Prof. Hermann Oberth in 1929 both described the advantages of using space electric rocket propulsion. Their ideas remained undeveloped because of the lack of a lightweight electric power system. When World War II brought the possibility of controlled nuclear fission, two English scientists, Dr. L. R. Shepherd and Mr. A. V. Cleaver, suggested in 1948 that nuclear electric power could provide the lightweight electric power source needed for space electric propulsion. In 1954, Dr. Ernst Stuhlinger presented the first detailed analysis of electric rocket systems. Dr. Stuhlinger's papers stimulated the interest of many laboratories, and one of those, the

NASA Lewis Research Center, began its program in space electric propulsion in 1957. Many types of electric thruster devices were proposed and evaluated by these laboratories. Of these, the mercury-bombardment ion thruster emerged as having the high specific impulse, high efficiency, and long lifetime characteristics needed for future high-energy missions in space.

Much of the mission analysis done in the late 1950's and early 1960's assumed the use of nuclear electric powerplants with megawatt power levels and a thrust level of 100 newtons. A typical mission was a 3-year manned Mars exploration. By 1963, however, solar cell technology had produced lightweight, efficient solar cells, and a 500-kilogram unmanned spacecraft using tens of kilowatts of solar electric power (0.5-N thrust) became a viable interplanetary design concept. About this time, proposed synchronous-orbit spacecraft north-south stationkeeping requirements created a desire for thruster systems with a specific impulse greater than 300 seconds for long life and with low thrust for accurate control. These requirements could be met ideally by 100-watt electric thrusters producing several millinewtons of thrust at specific impulse of 1000 to 3000 seconds. These electric thruster requirements have remained essentially unchanged to the present. In the late 1970's, orbit raising of large cargos was defined as an additional electric propulsion mission requirement. Such missions might require up to 100 kilowatts of power and thrusts of the order of 5 newtons.

Mercury-Bombardment Ion Thruster Research

In 1959, Dr. Harold Kaufman devised and tested the first mercury-bombardment ion thruster. By 1960 the first 10-centimeter-diameter working model of an ion thruster had been demonstrated. This is shown as a starting block in figure 6. The balance of figure 6 indicates the major thruster designs and how they formed the basis for technological evolution over the next 20 years.

The first 10-centimeter-diameter (10 cm) laboratory thruster established that a propellant gas could be ionized in a discharge chamber and extracted into an ion beam. Basic design parameters were varied in the discharge chamber to reduce the power lost in making ions.

The first major component improvement was to replace the early wire accelerator grids with a parallel-plate, multi-drilled-hole grid set. This new grid design made possible higher total impulse and eliminated structural support problems with the wire grids. Thruster scaling relationships were established by testing in both smaller (5 cm) and larger (20 cm) diameter thrusters.

The early cathode designs (hot-filament type) had lifetimes of several hours, which was sufficient to meet the objectives of the Space Electric Rocket Test I (SERT I) flight (see the following section Space Tests for details). The 10-cm laboratory thruster was made mechanically adequate to meet launch vibrations, a short-term feed system was installed, and the SERT I flight was performed in 1964. The cathode lifetime of the SERT I thruster, however, was insufficient to meet the requirements of future space missions.

On SERT I the neutralizer cathode was immersed in the ion beam and subject to high erosion by the beam. The main cathode was eroded at a lower rate by the discharge ions, but still had only a 150-hour lifetime. The major innovation of a plasma-bridge, hollow-cathode neutralizer was suggested by Michael Sellen of TRW. This was then developed at Lewis into a long-life, power-efficient neutralization system. The hardware of the neutralizer could be located outside the ion beam (thus giving the required long life) and use a "plasma bridge" to transfer neutralizing electrons into the ion beam. The main cathode lifetime goal was met by the Lewis design of a thick oxide-coated cathode. The oxide-coated cathode had both long life and low power requirements but required refurbishment following each air exposure of the ion thruster. Midway in SERT II flight thruster development the technology of the neutralizer hollow cathode was used to design a main-discharge hollow cathode. This type of cathode with minor improvements had been used in every subsequent thruster to the present (1980).

A porous tungsten vaporizer design ideally met feed system requirements of liquid-vapor interface control, on-off valve, vaporization of liquid mercury to gas, and propellant mass flow control. A blowdown storage tank with a rubber bladder separating mercury and nitrogen pressurant completed the feed system design for SERT II.

In parallel with the SERT I and II development programs a research program used earlier scaling results to successfully design and test both a 50-cm thruster and a 150-cm thruster. These experimental thrusters, which contained 4 and 10 main cathodes, respectively, were tested at power levels of 50 and 250 kilowatts, respectively. Their intended use was for megawatt-level, nuclear electric propulsion space missions.

The 150-cm thruster was ground tested at 250 kilowatts power level (1 N thrust). Work on this large thruster was stopped in the mid-1960's when the National decision was made to stop developing large space nuclear electric powerplants.

The SERT II flight (see the section Space Tests) was successfully launched in 1970. From the SERT II technology emerged a split program of auxiliary

(5 cm) and primary (30 cm) propulsion thruster development. At this point the Hughes Research Laboratory was contracted to perform the industrial development of these thrusters. Much of the basic thruster work was still performed at Lewis in a joint effort with Hughes.

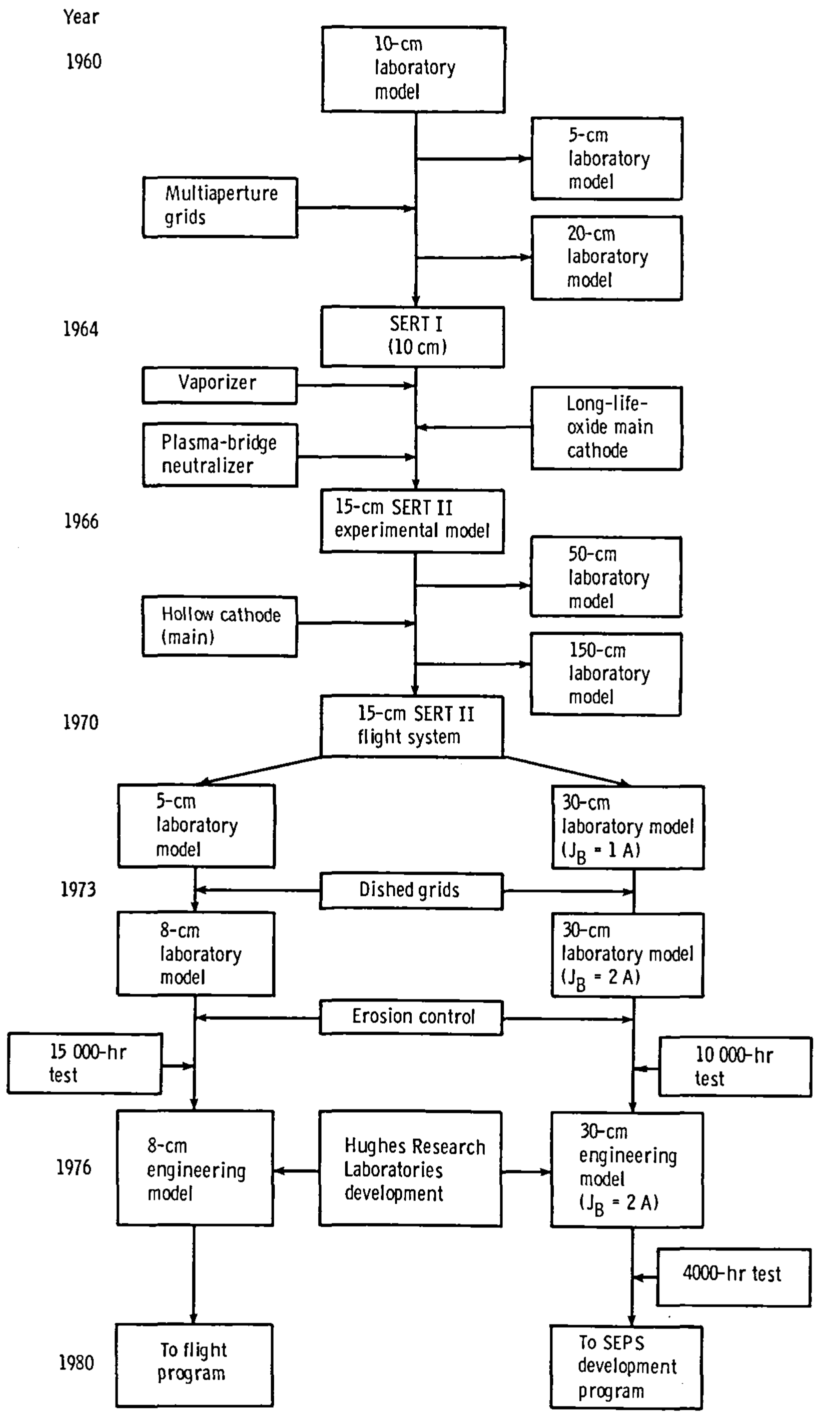


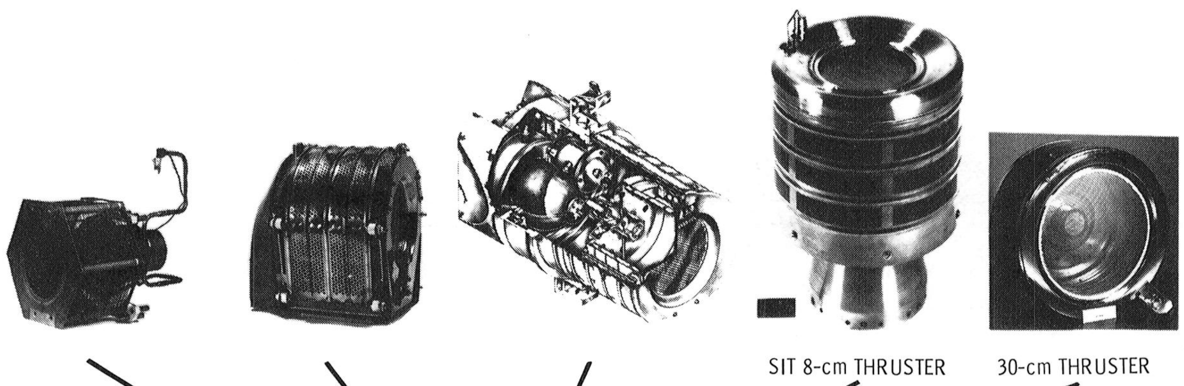
Figure 6.—History of mercury-bombardment ion thruster research.

Hughes invented a propellant flow electrical isolator, and structurally and thermally integrated it with hollow cathodes and vaporizers to make cathode isolator and neutralizer isolator vaporizers for the 5-cm thruster. Long-term thruster and component tests were performed to verify life. Erosion problems of the discharge chamber were solved by using erosion-resistant materials at critical locations, readjusting the operating conditions, and using wire-mesh surfaces to contain surface-deposited films. This technology was implemented in the 30-cm thruster after demonstration in the lower cost 5-cm thruster tests.

In 1968, the SERT II technology was incorporated into the preliminary designs of the present 30-cm Space Electric Propulsion System (SEPS) thruster. In 1971, the SERT II technology was used to design a 5-cm thruster and later the 8-cm thruster (fig. 7) that will be space tested on board the P80-1 spacecraft (the Teal Ruby) in the early 1980's. Both the 8-cm and 30-cm thruster systems have been developed to flight prototype hardware by Hughes. The 8-cm thruster system (150 W, 5 mN) is fully described in a users manual (NASA CR-162209, available on request from J. Pelouch, NASA Lewis Research Center), and its development history is summarized (ref. 2) by NASA's Office of Aeronautics and Space Technology.

The late 1960's decision to use a mercury-bombardment thruster design for primary propulsion was based on the following: (1) demonstrated operation at required power levels, (2) no major development or interface problems, and (3) equal or higher efficiency than competing thruster types. Further advantages were its ease of scaling for a range of missions and successful flight tests of early thrusters.

The choice of electric thruster type for auxiliary propulsion was not so clear cut as the choice for primary propulsion. Resistojets and Teflon, pulsed-plasma thrusters have been developed and are presently in use on spacecraft. However, both technologies lack the inherent high specific impulse and high total impulse of the bombardment ion thruster. Furthermore the ion thruster had demonstrated performance and life during both ground and space tests. Therefore NASA decided to develop this technology to flight demonstration and the Air Force Systems Command, in anticipation of its future needs, is supporting the test of this technology on the P80-1 satellite.



	1959 FIRST THRUSTER	1964 SERT I	1970 SERT II	1976 8-cm EM THRUSTER	1976 30-cm EM THRUSTER	1978 30-cm J SERIES THRUSTER
DIAM, cm THRUST, mN POWER, kW I_{sp}, sec	10 4 0.4 5000	10 25 1.5 4000	15 28 0.85 4200	8 5 0.125 2800	30 130 2.6 3000	30 130 2.7 3000
	LAB MODEL DEMO FEASIBILITY OF IDEA	31 min FLIGHT PROVED NEUT IN SPACE SHORT DESIGN LIFE	$5\frac{1}{2}$ MON FLT OPERATION 1ST USE OF HOLLOW CATH	MODULAR SIZES 2.5 TO 20 mN 5000 CYCLES DEMO	DISHED GRIDS LOWER I_{SP} GIVES HIGH T/P LARGER T, AT SAME P LOSS	LIFETIME IMPROVEMENTS INCORPORATED "SMALL HOLE" ACCEL GRID

Figure 7. – Mercury-bombardment ion thruster development at the Lewis Research Center.

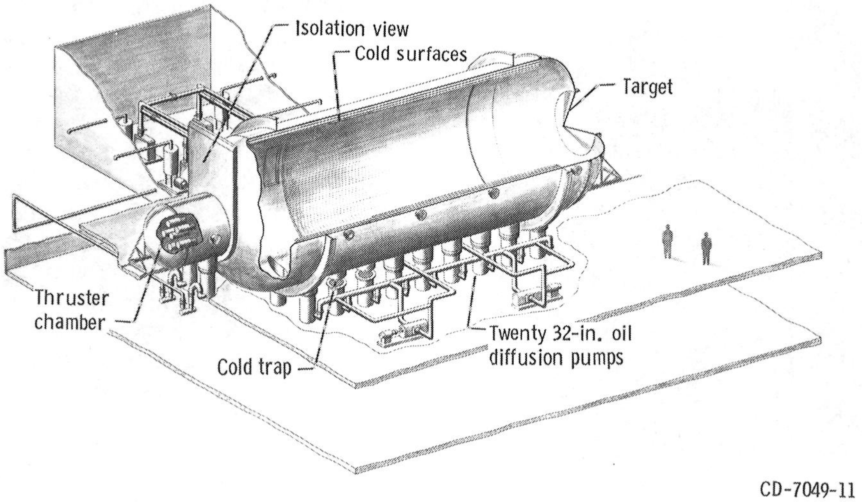


Figure 8. – Cutaway drawing of tank 6 in Electric Propulsion Laboratory.

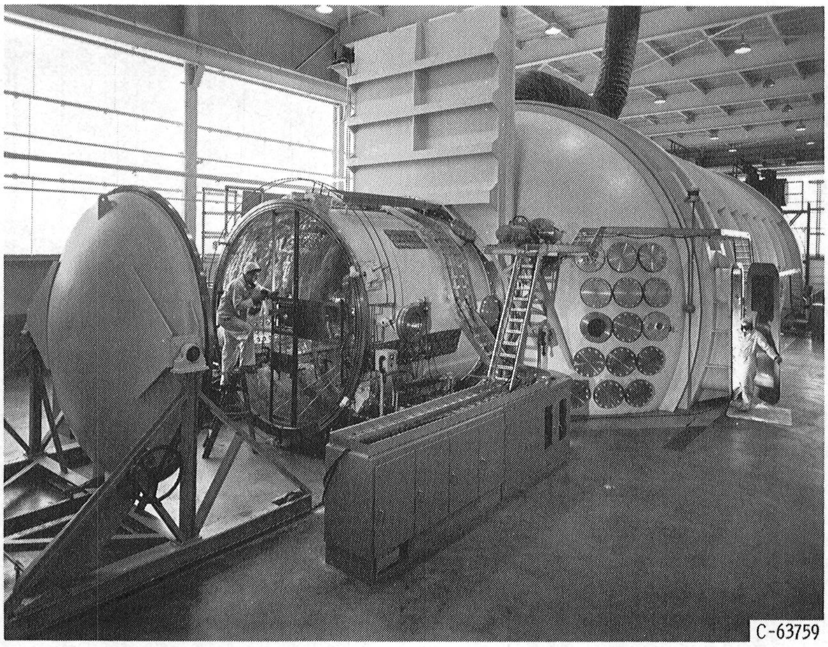


Figure 9. – Tank 6 in Electric Propulsion Laboratory.

Space Tests

SERT I. – In the early 1960's, the Lewis Research Center constructed two major space simulation facilities, the largest in the world designed solely for electric propulsion testing. These facilities, a 25-foot-diameter by 70-foot-long chamber designated tank 6 and a 15-foot-diameter by 70-foot-long chamber (tank 5), were housed in a laboratory building containing offices and ancillary support systems. This building is the Electric Propulsion Laboratory (EPL) at the Lewis Research Center (ref. 3).

Even though large vacuum tanks such as tanks 5 and 6 at EPL (tank 6 is shown in figs. 8 and 9) provide an excellent simulation of an environment for testing ion thrusters, some questions could be answered only by operating ion thrusters in space. One such question was how would the ions and electrons exhausting from the thruster interact with space plasma, when the walls of a vacuum tank would no longer surround the exhaust.

On July 20, 1964, two ion thrusters were briefly tested in space by NASA (ref. 4). One was a mercury-bombardment ion thruster (fig. 6) developed by the

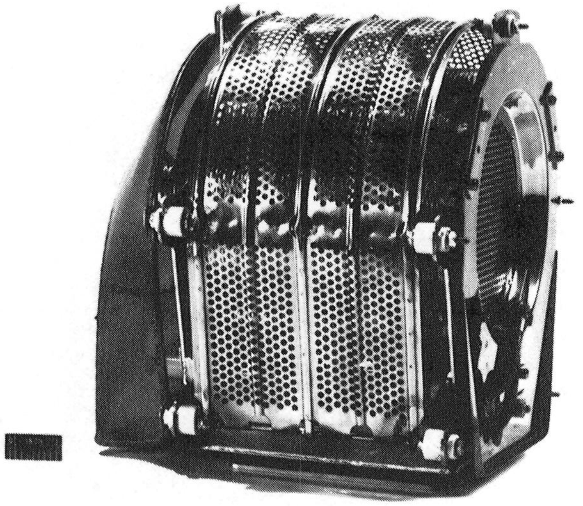


Figure 10. – SERT I ion thruster.

NASA Lewis Research Center; the other was a cesium-contact-ionization thruster developed by the Hughes Research Laboratories under NASA contract. This test was known as SERT I (Space Electric Rocket Test I). The battery-powered thrusters were mounted on a capsule that was launched with a Scout solid-propellant rocket into a ballistic trajectory (figs. 10 and 11).

These thrusters had been operated on the ground for hundreds of hours in vacuum tanks to measure their performance and qualify them for the space experiment. During these tests the cesium or mercury ions from the thrusters struck the walls of the vacuum tank and knocked many electrons loose from the walls (fig. 12). These electrons could have entered the exhaust beam and neutralized the ion space charge described previously. With the possibility of this neutralization occurring, it was difficult to tell whether the electrons from the thruster neutralizer were doing their job of neutralizing the ion beam.

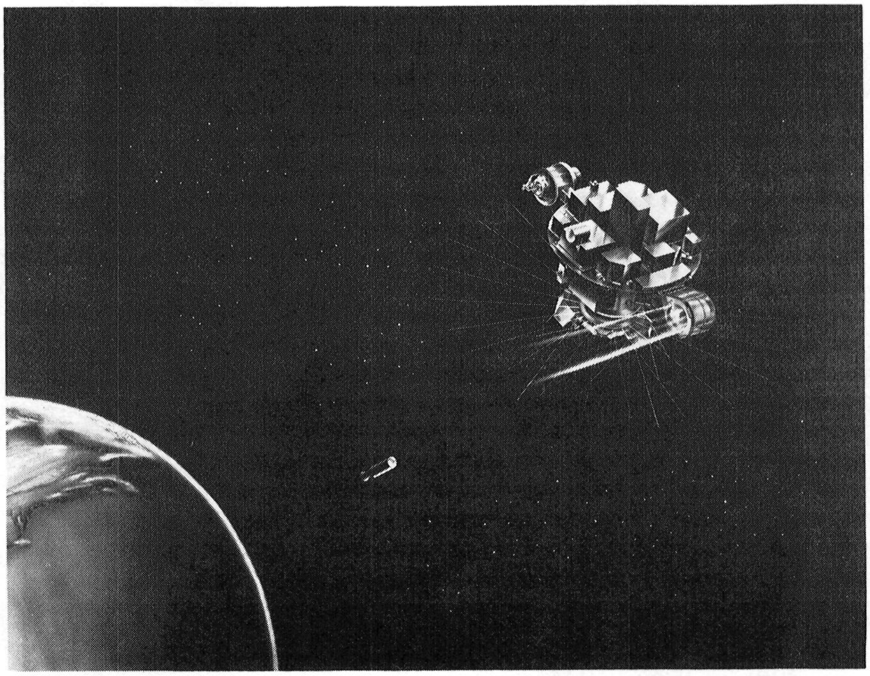


Figure 11. – SERT I in orbit.

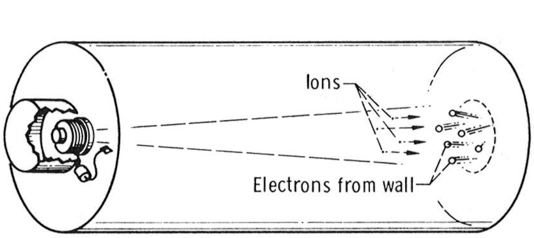


Figure 12. – Effect of ions on tank walls.

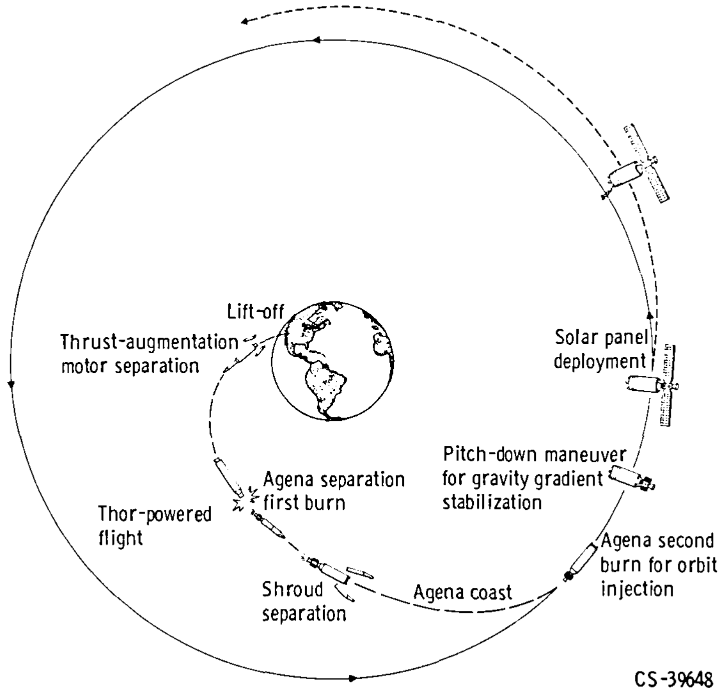


Figure 13. – Launch plan for SERT II.

The primary purpose of SERT I therefore was to demonstrate neutralization in space and to measure any differences between ground and space operation. The direct evidence of incorrect neutralization would be a decrease in thrust from the predicted values.

Because electric rocket thrusters have only small thrust, the 50-minute SERT I ballistic flight would not have been long enough for the thruster to change the spacecraft trajectory enough to obtain an accurate thrust measurement. Therefore the thrusters were mounted on arms so that their thrust would change the spin rate of the spin-stabilized spacecraft. The mercury-bombardment ion thruster (fig. 6) on board SERT I operated as predicted from vacuum chamber tests and produced thrust. Thus the very important process of neutralization in space was proved possible.

The cesium-contact-ionization thruster also on board SERT I did not operate because of a short in the high-voltage wiring, but on August 29, 1964, a cesium-contact-ionization thruster produced thrust in a similar space test conducted by the U.S. Air Force. The Air Force test thruster was developed by Electro-Optical Systems of Pasadena, California. Small electrothermal, resistojet thrusters were successfully tested in space in the late 1960's. Small ground-based tests in vacuum facilities. The next step was to determine how durable the thrusters were.

SERT II. – The SERT I flight verified the neutralization of an ion beam in space by showing the production of thrust, but it was a short flight that used batteries for power. The purpose of the SERT II flight was to demonstrate long-term operation of an ion thruster in space with a flight type of power source.

On February 3, 1970, the SERT II spacecraft was launched by a Thor-Agena launch vehicle into a circular polar orbit, as shown in figure 13. The polar orbit permitted the solar panels that powered the thruster to remain in continuous sunlight throughout an entire orbit. The solar panels, each 1.5 by 5.8 meters, are shown in figure 14 together with the complete spacecraft built onto the Agena stage.

Figure 15 shows one of the two SERT II thrusters. Each thruster was a 15-centimeter-diameter mercury-bombardment ion thruster and at full power used 850 watts to produce 28 millinewtons of thrust. The thruster was also able to operate at 40 and 80 percent of full power.

The SERT II flight provided 5 months of successful operation with one thruster and 3 months with the other thruster (ref. 5). A minor thruster redesign as a result of follow-on ground tests (ref. 6) has extended the design lifetime to 15 to 30 months. Ground life tests of identical thruster power-processor systems were stopped without failure after 9 and 8 months, respectively, of continuous running. The thrust of the SERT II ion thruster was measured by an onboard accelerometer and by the change in the spacecraft's orbit. These two measurements of

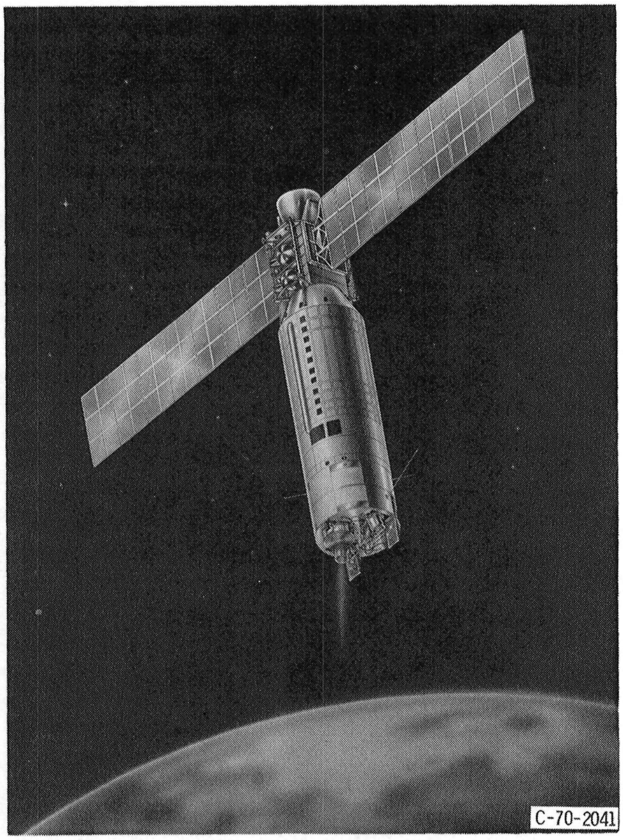


Figure 14. – SERT II in orbit.

thrust and a thrust calculated from measured beam current and voltage produced identical values of thrust within experimental error.

Other experiments successfully performed were (1) a neutralizer cathode bias experiment in which the entire spacecraft was biased from 50 volts negative to 8 volts positive of space plasma; (2) probe sweeping of the ion beam, which showed a beam profile in space similar to those measured in ground tests (small beam divergence loss); and (3) a radiofrequency-interference measurement, which indicated no radiofrequency-interference ion beam noise detectable above the background noise level of Earth and thus gave evidence that stations on Earth will be able to communicate with future spacecraft without interference from thruster ion beam plasma noise.

The SERT II spacecraft and thruster system components have been operated periodically for over 10 years in space. Short (1 hr) periods of sunlight on the SERT II solar arrays permitted brief testing of the thruster systems and components from 1973 to 1978. In 1979 and 1980 the spacecraft entered an orbit of continuous sunlight and gave opportunity to run continuous thruster tests.

The high-voltage grid short that limited thruster lifetime in 1970 was cleared from one thruster system in 1974. This thruster system then functioned normally in short tests (1974–78) and operated for 1700 hours in 1979–80 until its neutralizer propellant

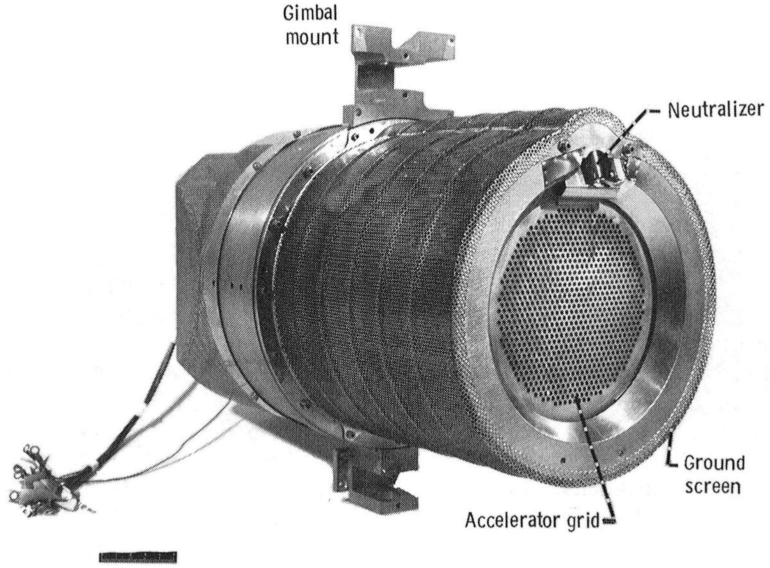


Figure 15. – SERT II ion thruster.

reservoir was depleted as expected. Following this depletion the thruster was again operated by using the neutralizer system from the other thruster system. This cross-neutralization from another neutralizer 1 meter away was an unplanned result and has important design influences on neutralizer operation of future spacecraft containing an array of ion thrusters (ref. 7). Also in 1979, the ion thrusters were used for the first time to aid in the spacecraft attitude control. The ion thrusters were gimbaled to thrust off the spacecraft center of gravity. The torque thus produced increased the spin rate of the spacecraft to maintain stability. The torque value was reduced to a thrust value that was the same value as measured in 1970 by orbit charge and by the onboard accelerometer.

In late 1979 and 1980 both SERT II thrusters were operated in a "plasma thrust" mode. In this mode the high voltage to the accelerator grids was turned off, but the main discharge was left on. Mercury ions were pushed out through the grids by the discharge voltage. Equal numbers of electrons were pulled along by the ions and a neutral plasma beam emerged. The thrust of this plasma beam was measured by change in spacecraft spin rate and found to be 0.8 millinewton. The thrust so produced was much less than the normal ion thrust (28 mN), but still enough to affect (even control) the attitude of a SEPS spacecraft.

Two propellant flow systems and two hollow cathodes on each of the two flight thrusters have been started successfully over 200 times in space. Storage periods between starts ranged from 10 minutes to 350 days. Each thruster power processor has operated without incident for over 4000 hours in space during the 10-year period (ref. 8). The spacecraft solar arrays and thermal control surfaces have shown no abnormal degradation due to contamination from thruster operation. The spacecraft attitude-control system also continues to function correctly.

Continued operation of each thruster system main-discharge chamber is planned to obtain further confidence of the hollow-cathode discharge design and propellant flow systems. The main propellant reservoir on one system was exhausted in December 1980, and the other is projected to be exhausted in May 1981. Continuous sunlight will be lost after May 1981 (except for a 2 months in late 1981) for 7 years.

The SERT II flight has provided mission planners with important data needed to design space electric propulsion systems for long-duration missions. These data, combined with data from over 200 000 hours of thruster ground tests, provide a confident basis for evaluating thruster operating lifetime and thrust performance level and for designing future spacecraft so as to avoid contamination.

Power Processing

Ion thruster systems need to have the power from the spacecraft bus conditioned for their particular requirements. Ion thrusters in particular require power processing to generate, control, accelerate, and neutralize the ion beam.

The power-processing equipment must perform matching between the electric thruster and the power source, not only to meet the steady power requirements, but also to handle the transients that arise through engine operation. These include the consequences of operating under fixed conditions by means of control circuits performing dynamic control, the transients that arise through the special demands of the thruster to change operating conditions in performing the particular mission, and the unsteady operation that arises through the employment of a stop-restart cycle that is switched in to quench serious arcing in the interelectrode space.

The primary functions that the power processing equipment may have to perform are (1) dc to ac conversion, (2) ac to ac change of voltage level, (3) ac to dc rectification, (4) regulation of voltage, (5) regulation of current, (6) regulation of power, (7) protection against power surge, and (8) distribution of power. These functions are clearly not separable from the control functions that must be performed to regulate thrust (beam current and voltage), percentage of ionization (mass flow and arc discharge conditions), and specific impulse (discharge chamber voltage).

In addition, to a large extent, the power processor unit (PPU) must be autonomous: It must be capable of maintaining system health with only occasional updates to system operating conditions. It must be efficient since every percentage of power loss must be made up for with additional solar array area, as well as increased mass for the radiation rejection of heat. Finally, it must be lightweight and structurally capable of withstanding the launch environment.

SERT I. – The power processing for the 10-cm mercury ion thruster suborbital flight (SERT I) that was flown in 1964 was developed by the TRW Electromechanical Division in Cleveland, Ohio (ref. 9).

Power for the 10-cm mercury ion thruster was provided by a power converter that obtained power from a 56-volt silver-zinc battery. Two additional batteries were used to provide power to the thruster's magnetic field coil and the neutralizer filament directly. Because of the high voltage level upon which the direct-feed magnetic field and neutralizer batteries floated, these assemblies consisted of sealed cells housed in fiberglass cases.

The five electronic supplies for the mercury ion thruster were assembled into two cases, two high-

voltage supplies in one and three low-voltage supplies in the other. The two cases weighed 60 pounds and developed 1.335 kilowatts at an efficiency of 85 percent. The cases were designed structurally to add to the structural integrity of the SERT I spacecraft baseplate. The cases weighed 27 pounds and the electronics 33 pounds.

The SERT I power processor operated at a 1-kilohertz switching speed because of the switching speed limitations of available power transistors at that time. The transformers contributed a large percentage of the electronics weight as a result of the low switching speed.

To eliminate concern over voltage breakdown due to outgassing during the short suborbital flight, the power supplies were sealed and pressurized with nitrogen.

The SERT I power processors for the mercury ion thrusters performed without incident during the SERT I flight.

SERT II. – The power processing development program for SERT II incorporated concepts that proved valuable in assuring its long-term performance for a 6-month mission operating an electric thruster. The power-processing system described here was developed to power a 1-kilowatt mercury-bombardment ion thruster on the orbital SERT II spacecraft by Westinghouse Corp. under NASA contract (ref. 10). The power-processing system used for the 1964 SERT I mission incorporated a sealed, pressurized enclosure to ensure proper operation during its 1-hour useful mission life. The SERT II system was specifically designed to perform a 6-month mission.

The information gained from the development and application of this power processor covered several design areas, including protection of the power conditioner from both external and internal arcs and open-to-vacuum construction. Late in the development program, failures due to internal arcing began occurring. This was particularly frustrating because of the precautions already taken. The subsequent investigation showed that insulation was required inside the power processor. Installing this insulation solved the breakdown problem.

The power processor unit (PPU) converts solar cell electric power into power required to operate a mercury-bombardment ion thruster. The SERT II solar cell array provides a nominal dc voltage output of 60 volts during full-beam thruster operation. The PPU converts this into nine different electrical outputs totaling approximately 860 watts for the thruster. The major amount of electric power was delivered at 3000 volts and 0.25 ampere dc. Nominal conversion efficiency was 87 percent.

The PPU with and without cover is shown in figures 16 and 17. It is a rectangular box 52 centimeters (20.5 in.) long, 26.7 centimeters (10.5 in.) wide, and 14 centimeters (5.5 in.) high and weighs 14.5 kilograms (32 lb). The PPU is bolted to the spacecraft radiator. The radiator is sized to maintain a temperature below 120° F at rated operating conditions.

The PPU performance in space has been perfect without exception. All supplies powered their thruster loads well within their specified ranges. Thruster arcs occurred on an average of four per day, and the PPU safely handled overloads with no apparent adverse effects. All control loops were stable, and the measured operating parameters were constant within the telemetry resolution.

Flight performance compares very well with ground testing. All operating values equal those obtained during ground testing, where the PPU and thruster were mounted in their flight configuration on the spacecraft in a vacuum tank. Although certain test conditions on the ground resulted in more frequent thruster arcs, the arcs during long uninterrupted tests compare favorably with the number recorded in flight.

Thirty-centimeter-diameter ion thruster power processor development. – Three 30-cm thruster power processor breadboard programs existed between 1972 and early 1975: the Hughes thermal vacuum breadboard (TVBB), the TRW TVBB, and the TRW three-inverter breadboard. The Hughes TVBB design, which uses a uniquely driven, transistor, full-bridge inverter power stage, was tested on a 30-cm ion thruster in both ambient and thermal vacuum environments. The TRW TVBB and three-inverter breadboard use series-resonant, thyristor (silicon-controlled rectifier, SCR), half-bridge inverter power stages. Both of these units underwent thruster integration testing, and the TRW TVBB saw limited thermal vacuum operation. In February 1975, a management decision was made to pursue the series-resonant SCR inverter power processor as the baseline design.

To date, the TRW TVBB and three-inverter power processors have accrued over 20 000 hours of thruster operation (with no power component failures). During this time these two power processors were used extensively for thruster-power processor development and characterization and for numerous parametric, optimization, interaction, and control studies. The results of this work established the thruster-power processor voltage, current, set points, and interface requirements; the high-voltage recycle procedure; and the critical interrupt parameters and their magnitudes. It also established the thruster startup, throttle, and shutdown algorithms and the control requirements, particularly the main, cathode, and neutralizer control loops. These data have been incorporated in the functional-

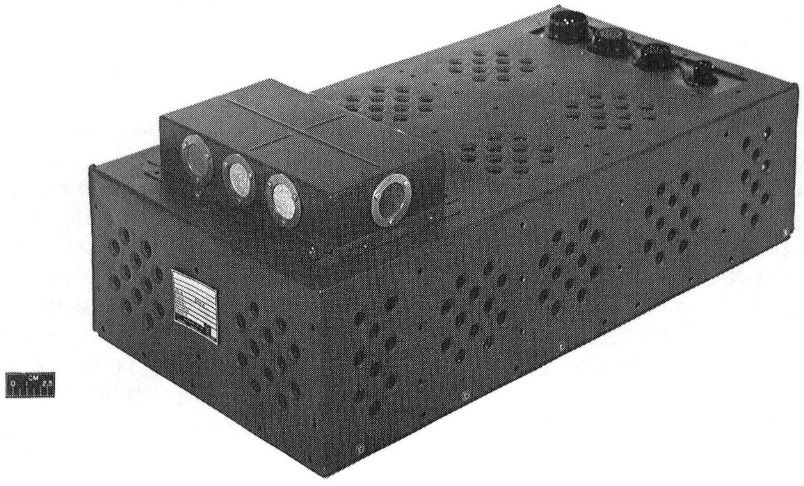


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Figure 16. – Power processor with cover installed.

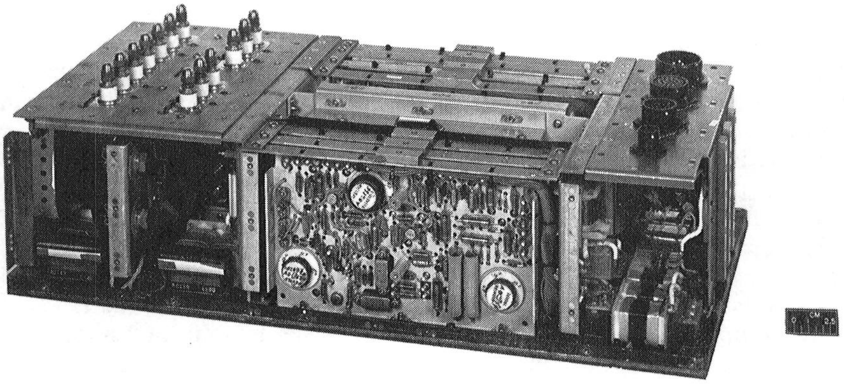

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Figure 17. – Power processor without cover.

model power processor unit (FM/PPU) power processor electrical design. Also, several minor electrical design changes have occurred as a result of FM/PPU power processor testing.

The electrical-prototype power processing unit (EP/PPU) is an electrical brassboard produced under a Lewis contracted effort with TRW Defense and Space Systems Group (ref. 11) from mid-1975

through 1976. The objective of the EP/PPU effort was to design, fabricate, and test a power processor that would meet electrical specifications typical of a flight mission. The EP/PPU was designed to incorporate an integrated modular packaging approach, although no detailed structural or thermal flight analysis was performed under this contract. The subsequent FM/PPU effort used the EP/PPU electrical design and further refined the packaging by optimizing packaging density.

The EP/PPU contract also provided the electronic parts (commercial equivalents of high-reliability parts) and flight magnetics sufficient to fabricate five FM/PPU's. The EP/PPU has undergone several thousand hours of thruster testing, including vacuum operation with a thruster. Electromagnetic interference (EMI) tests were performed on the EP/PPU while it was operating a thruster, and radiated emission tests were performed with a thruster load bank. Figure 18 is a block diagram of the PPU.

The baseline Lewis FM/PPU is the most up-to-date power processor hardware available. The electrical design of the FM/PPU is identical to that of the EP/PPU. The FM/PPU was designed for flight structural, thermal, and environmental requirements (ref. 12). Figure 19 shows the FM/PPU with the covers removed.

From 1972 to 1974, a number of FM/PPU packaging concepts were considered in the course of the Advanced Systems Technology (AST) program with the Jet Propulsion Laboratory, the Marshall Space Flight Center, and the Lewis Research Center. Most of these concepts used direct radiation to space through louvers for thermal control. The two earliest competing concepts had the electrical components either bolted directly to the radiators or bolted to the webs of the crossbeams, sandwiched between two plates. A study conducted in 1974 showed that an all-heat-pipe thermal control system would be most weight efficient for the large heat dissipations anticipated (ref. 13). The dual shear plate approach was then modified by enlarging one flange of the crossbeam enough to accommodate the heavy or high-heat-dissipating components. The flange was then bolted directly to the heat pipe saddles. That approach eventually evolved into a packaging arrangement for two PPU's bolted to a common redundant heat pipe system in order to further reduce structural and thermal control system mass.

Five FM/PPU's have been fabricated at Lewis. Thermal vacuum qualification and vibration qualification testing has been successfully completed. Long-term vacuum testing in excess of 3000 hours was completed on one unit. These five PPU's have accumulated thousands of hours in support of systems testing at Lewis and in mission profile life testing.

Since the FM/PPU design there have been two contracts to investigate techniques to further the baseline FM/PPU performance. The first contract, the Extended Performance Electric Propulsion Power Processor Design Study, compared and evaluated concepts for improving the performance and increasing the power output of the power processor to as high as 10 kilowatts. This study, which was begun to support a possible 1985 Halley's comet rendezvous mission, was conducted by TRW from May to October 1977 (ref. 14).

The most recent contracted effort was the Improved Power Processor Design Study (ref. 15). This effort furthered the designs and data base in the extended-performance design study. This was done by incorporating updated thruster requirements to optimize power supply sizing, performing trade-off studies, and updating electrical components (e.g., complementary metal oxide semiconductor logic). It also included incorporating a microprocessor, refining the power stage and control logic, performing stability analysis, and performing a power processor reliability assessment. This contract, which started in October 1978 with TRW, extended until 1980 and produced an updated electrical brassboard power processor.

Advanced Systems Technology Program

After the SERT II flight in 1970, the Lewis Research Center recognized and corrected the SERT II thruster high-voltage-grid short anomaly and by 1972 had tested prototypes of a 2.5-kilowatt, 30-cm ion thruster module with a capability in excess of 1 million pound-seconds total impulse at a specific impulse of 3000 seconds.

The capabilities of the 30-cm ion thruster, when incorporated in a solar electric propulsion (SEP) system, was enabling for large classes of previously unattainable planetary missions. The OAST-Lewis goal was to have technology readiness for this system by the end of fiscal year 1978.

The Jet Propulsion Laboratory planetary scientists had long desired to effect a rendezvous with an active comet as it achieved perihelion. Comets represent agglomerations of the primordial matter from which all the solar system is composed. One comet, Encke, was identified as being especially desirable for study. Encke is a young comet and thus has not been depleted of its volatile constituents. It is a short-period comet, returning approximately every 3 years. At perihelion it comes within 0.3 AU of the Sun, its coma is large, and the particulate density is high and relatively easy to measure. Encke has been studied

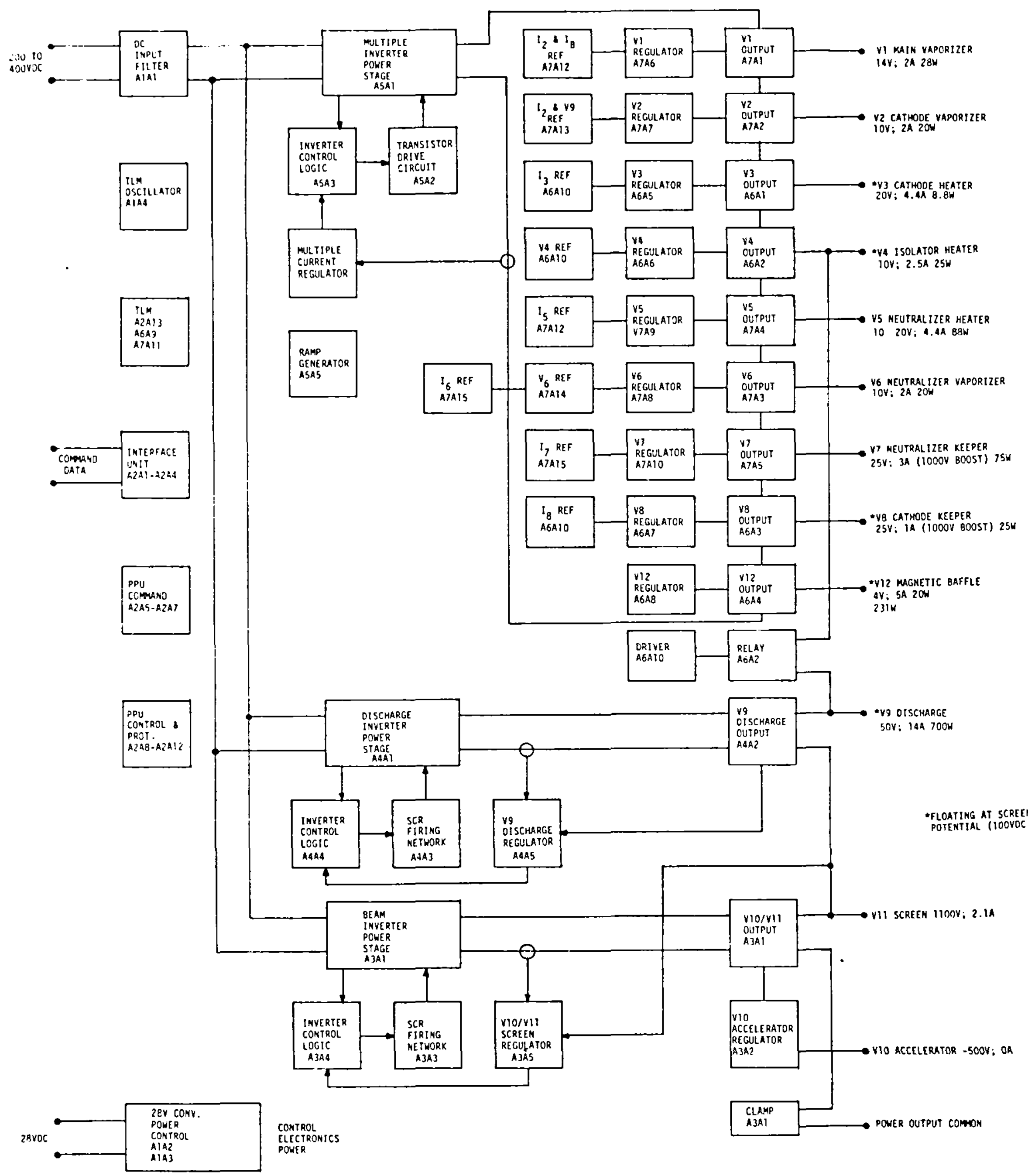


Figure 18.—Block diagram of power processor.

extensively by astronomers and its ephemeris is well known. A rendezvous with Encke, that is, a close approach where the relative velocities are matched within 4 meters per second at a distance of only a few kilometers, could provide an opportunity to garner inestimable amounts of scientific data.

From a scientific standpoint, 1984 should be an especially good apparition of Encke. During the next opportunity, in 1987, perihelion will occur when the Sun is between the Earth and the comet, precluding any data reception during the most interesting phase of the mission.

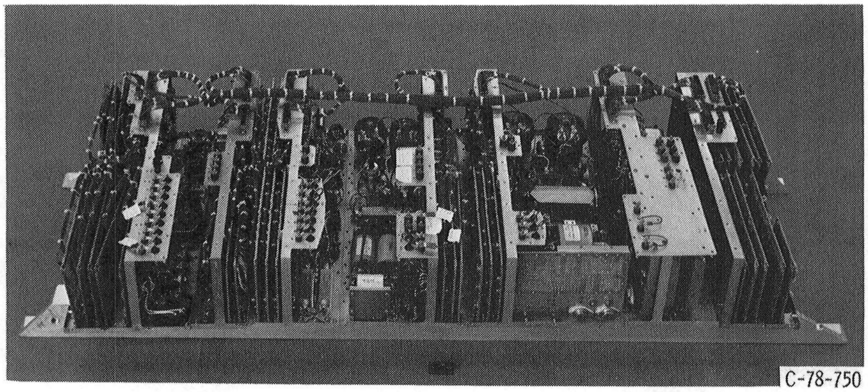


Figure 19.—Assembly of functional-model power processor unit.

Because of the extreme difficulty of the rendezvous mission it was decided to first attempt a precursor slow-flyby mission during the 1981 apparition and then a rendezvous during the 1984 apparition. The slow flyby only requires matching relative velocities to within 4 to 8 kilometers per second. The scientific yield from such a mission would, however, be correspondingly less than from a rendevous. Both missions require that the spacecraft be launched outbound in the solar system, with an aphelion near the orbit of Jupiter. The spacecraft would then attempt to come close to and match the velocity of Encke as they accelerated toward the Sun. Both missions require 2 to 3 years from launch to encounter with the comet. The slow flyby required an early 1979 launch, which was inconsistent with the technology readiness date of 1978 set by OAST and Lewis. Neither mission could be accomplished without the capabilities provided by a SEP system.

On June 1, 1973, Dr. John E. Naugle, Associate Administrator for the Office of Space Science (OSS) made a formal request to Roy P. Jackson, Associate Administrator for the Office of Aeronautics and Space Technology (OAST), to accelerate the 30-cm ion thruster system technology readiness date from 1978 to 1976. The request aroused some concern at Lewis. Acceleration of the program would mandate that a single set of performance requirements and system interfaces be established as early as possible. Lewis was at that time receiving conflicting sets of requirements from the Marshall Space Flight Center (MSFC), who were studying a solar electric propulsion stage (SEPS) for Earth-orbital operations, and the Jet Propulsion Laboratory (JPL) with their planetary system requirements.

On November 5, 1973, a letter from NASA Headquarters was sent to the Center Directors at Lewis, MSFC, and JPL establishing an Advanced Systems Technology (AST) program. The objective of the AST was to establish a single set of technical requirements for solar electric propulsion that would satisfy both planetary and Earth-orbital missions. Center responsibilities as defined in this letter were as follows:

For MSFC:

(1) Establish SEP concept requirements, definition, and design under the direction and guidance of NASA Headquarters with the objective of supporting a Comet Encke slow-flyby mission launched in December 1978 or January 1979 while still being applicable to a SEPS

(2) Coordinate the SEP AST activity to assure that the single set of technology requirements are met

(3) Develop appropriate solar array technology

For JPL:

(1) Demonstrate, through the AST program, technology readiness for mission use of the integrated-thrust subsystem

(2) Establish the Comet Encke slow-flyby mission definition, spacecraft system design, and requirements

For Lewis: Develop thruster and power conditioner technology and furnish necessary hardware to be used in the AST program

After the formation of the AST program, regular meetings were held at the participating Centers, working groups were established, and a set of interface control documents was developed. These documents established a referenceable set of performance requirements from which an ion thruster system could be built and tested.

In 1974, when it became apparent that the NASA budget would not allow project starts in the planetary programs in 1976, the SEP-related efforts at MSFC and JPL were virtually disbanded. The technology readiness program at Lewis was renegotiated with OAST and stretched to complete in 1980, maintaining the same AST requirements as the design criteria.

Focusing of Technology for Primary Electric Propulsion

A perspective of the technology readiness program in relation to mission programs that were seriously considered in recent years is shown in figure 20. A legacy from the AST program was that thruster and power processor performance requirements had been established and were used throughout the technology readiness program.

In 1977, a mission was proposed to develop a space vehicle to rendezvous with Halley's comet, which returns in 1986 after 75 years absence. This was not only an exciting scientific mission, but also had potential for public appeal that would improve its chances for approval. The technology readiness program formed a base—a departure point—for those mission studies. Even though the Halley's comet rendezvous mission was not approved, the technology readiness program continued.

Once it became too late to implement a mission to rendezvous with the Halley's comet, a mission was proposed to fly by Halley's comet and rendezvous with Tempel II comet. These studies also used the technology readiness program as a departure point. This mission is still under consideration. In March 1979, NASA made a commitment to develop a solar electric propulsion stage, and MSFC was chosen as the developing center. Competitive phase B studies are now under way to develop a general-purpose SEPS to perform the Halley flyby/Tempel II rendezvous mission and five other missions in the mission set. The technology readiness program continues toward its conclusion. The configuration of hardware that has evolved over the years has not been changed to coincide with the specific designs being developed by mission planners. In spite of this, the hardware of the technology readiness program is "flight like" enough so that most test results and experience are relevant to planned space vehicles. Keeping the hardware configurations fixed allowed the program to proceed to an orderly conclusion, whereas continuous hardware configuration changes would have greatly increased the cost and delayed the schedule with little benefit to demonstrating the readiness of the technology.

After more than two decades of research, electric propulsion has reached an advanced state of understanding. NASA has committed to develop an electrically propelled upper stage launch vehicle. A technology readiness program for primary electric propulsion completed in 1980 demonstrated that this technology is mature and ready to be developed into reliable space hardware. The dividing line between the completion of technology readiness and the start of development is not extremely sharp. For that reason, although the technology readiness program provided enough data by the end of 1980 so that development effort can be started with confidence, testing continues in 1981 to complete some of the long-term tests of the technology readiness program.

Technology Readiness Program

The technology readiness program described in this section is based on the 30-centimeter-diameter mercury ion thruster, which operates at a specific impulse as high as 3000 seconds and a beam current up to 2.0 amperes (fig. 21). Since the end use of the technology was application to a wide variety of planetary and Earth-orbital missions rather than to a specific mission, objectives and requirements were necessarily general and the concepts developed have wide-ranging applications. The stated objective of

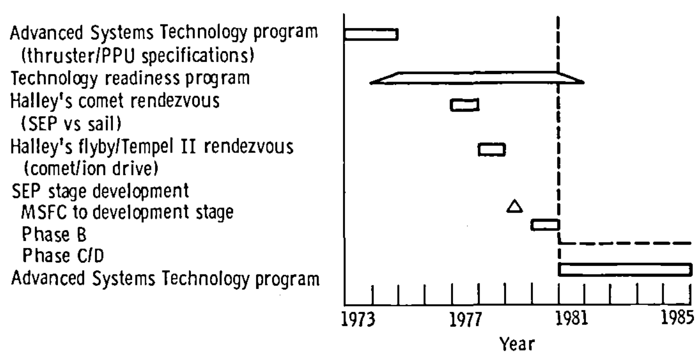


Figure 20.—Perspective of electric propulsion technology readiness program.

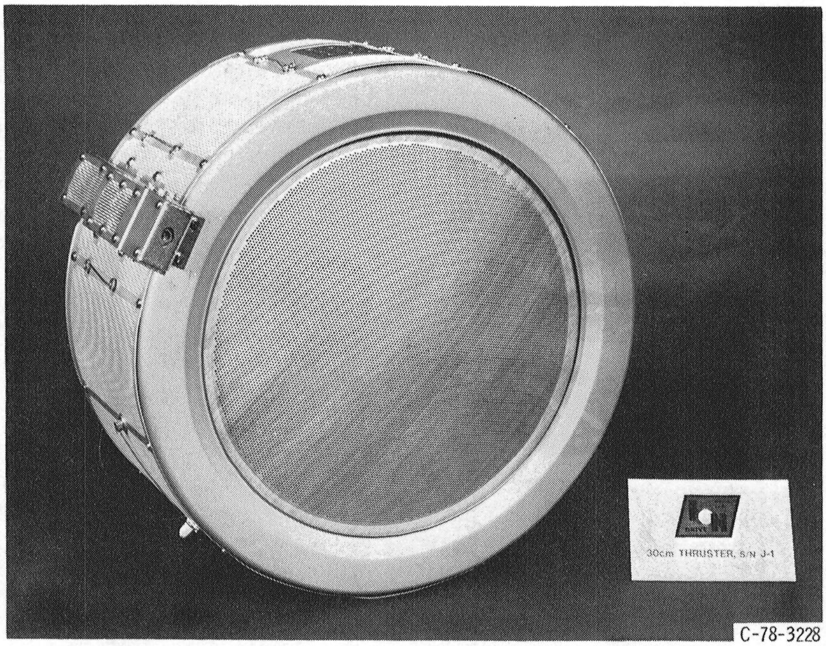


Figure 21. – Ion thruster—exhaust side.

the technology readiness program is to provide and demonstrate by the end of fiscal year 1980 the technology for primary solar electric propulsion ion thrust subsystems (thrusters, power processors, propellant system, thrust vector system, thermal control system) capable of operating over a wide solar power profile range such as would be encountered in Earth-orbital and planetary missions. The definition of technology readiness that applies to this program is that critical engineering problems will be identified and solutions provided such that system performance, interface requirements, and constraints will be defined to the point where application of the technology can be accomplished with known and acceptable risk.

Although the program was conceived to develop and evaluate technology, it was recognized that a system configuration concept was necessary in order to focus the technology effort and to provide guidance for decisions on configuring subsystem elements. Existence of a system concept would ensure that the elements of the subsystem once developed would be compatible and truly integrable into a functional space system. The concept used to focus this technology is described here so that a better understanding of the technology readiness program can result.

System concept. – Figure 22 shows one configuration of an electrically propelled spacecraft. Three major elements of this spacecraft are the mission-science modules, the solar array, and the ion propulsion system. The ion propulsion system, shown in figure 23, is an example of the modularity that can be installed. The building block is named the "bimod" and consists of two thrusters and two power processors with the heat pipe and radiator thermal control system. The ion propulsion system can be a collection of several "bimods" joined at the forward end by an interface truss. This interface truss could contain hardware such as propellant tankage and feed systems, computers or controllers, and the mission-specific hardware required. The technology readiness program concentrated on building and testing one bimod to demonstrate the feasibility of the approach. Also, some critical technology items needed in the interface truss were designed and built. To understand the hardware configurations discussed later, it is necessary to understand the bimod thermal control system.

The ion thrusters are cooled by direct radiation to space. However, the heat from the power processors must be rejected by radiators. The power processor design is divided into seven modules—each containing a set of compatible functions needed to operate an ion thruster (fig. 24). The power processor modules are mounted to heat pipe evaporator modules as shown in figure 25.

Each module was designed with the high-heat-dissipating components mounted to the baseplate so that the shortest possible heat path and lowest temperature drop could be maintained. One power supply was mounted to each side of the heat pipe evaporator saddle. Six variable-conductance heat pipes are required for each bimod. Three heat pipes transport heat to each radiator. The variable-conductance feature is needed so that, when the

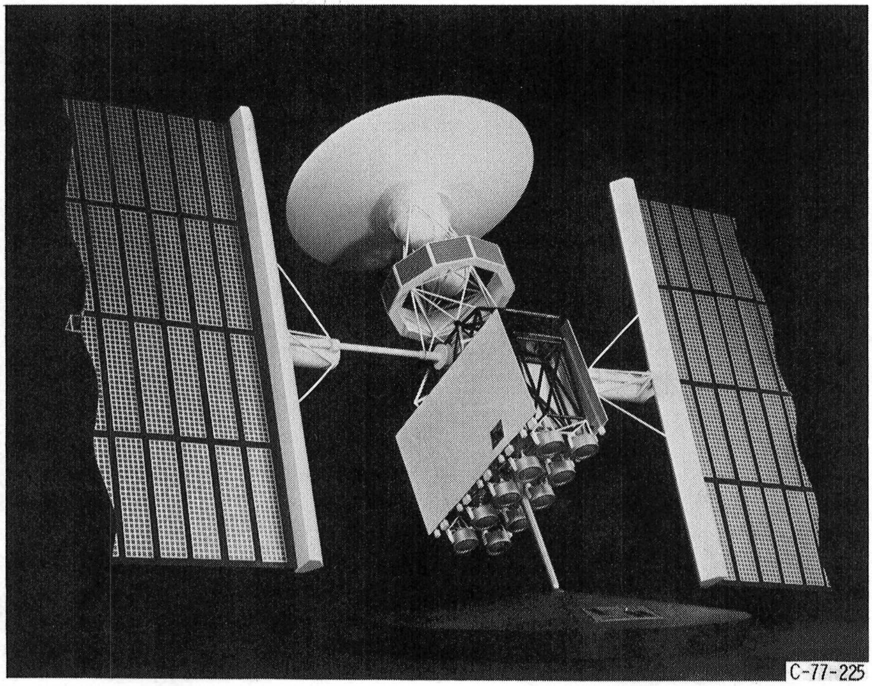


Figure 22. — One configuration of an electrically propelled spacecraft.

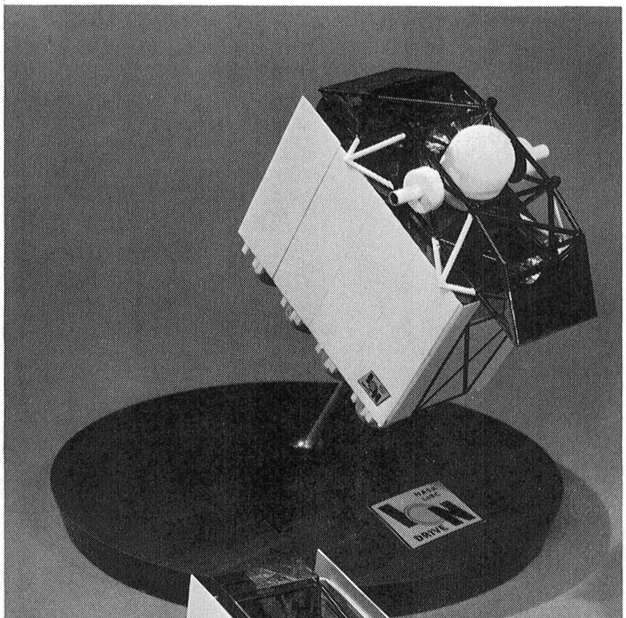


Figure 23. — Thrust system with bimod thrust module removed.

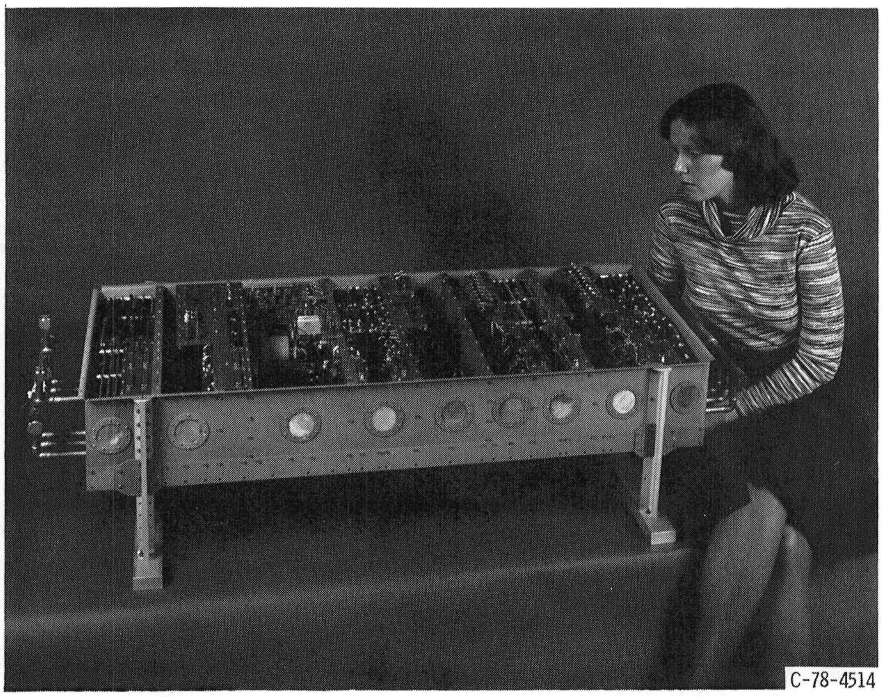


Figure 24. – Power processor.

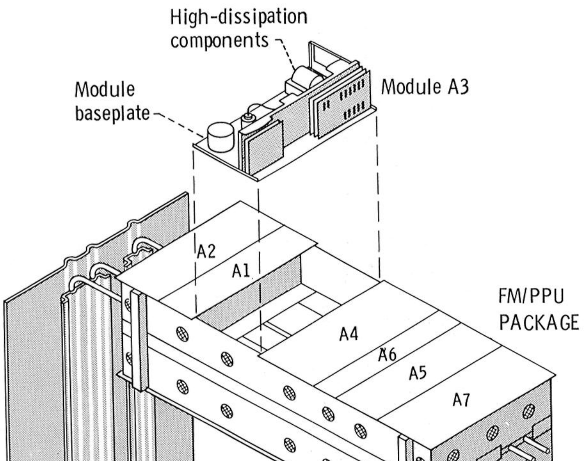


Figure 25. – Thermal control system.

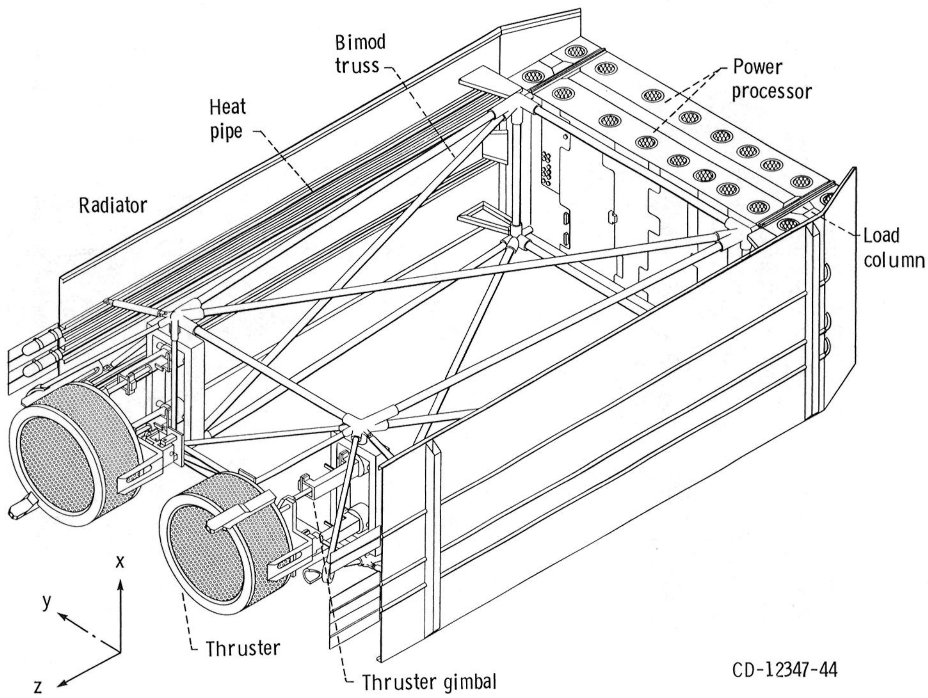
power processors are turned off, conduction to the radiators stops and thus the PPU's are not subjected to too low a temperature range. Heaters could be added if required, but that is a solar array load to be avoided if possible. Figure 26 is a line drawing of the bimod engine system showing its construction details.

Program elements. – The technology readiness program was constrained and guided by the objective, definition, and system concept just discussed. The program required development, testing, and evaluation of several elements (fig. 27). Thrusters, power processors, heat pipes, gimbals, propellant storage and distribution, and the high-power switch each underwent test and evaluation programs separate from the system level evaluation programs. The bimod test facility and bimod's mounted for testing are shown in figures 28 and 29.

Details of the elements and their functional tests and interface requirements are given in reference 11.

Technology Transfer Program

Resources. – Electric propulsion has always been a strong in-house program at the Lewis Research Center. This in-house expertise started with the initial demonstration of the electron-bombardment thruster and continued through the overall technological development of the 30-cm thruster subsystem. All SERT I and SERT II thruster design, fabrication, integration testing, and systems testing were done at



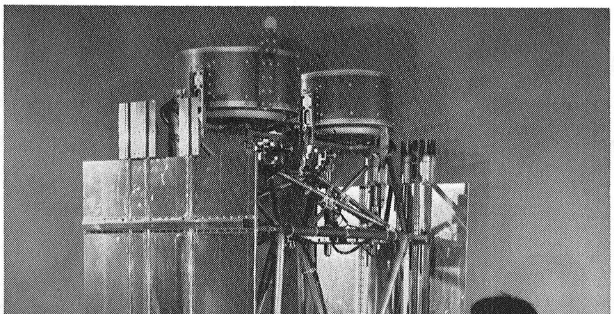


Figure 26. – Bimod engine system.

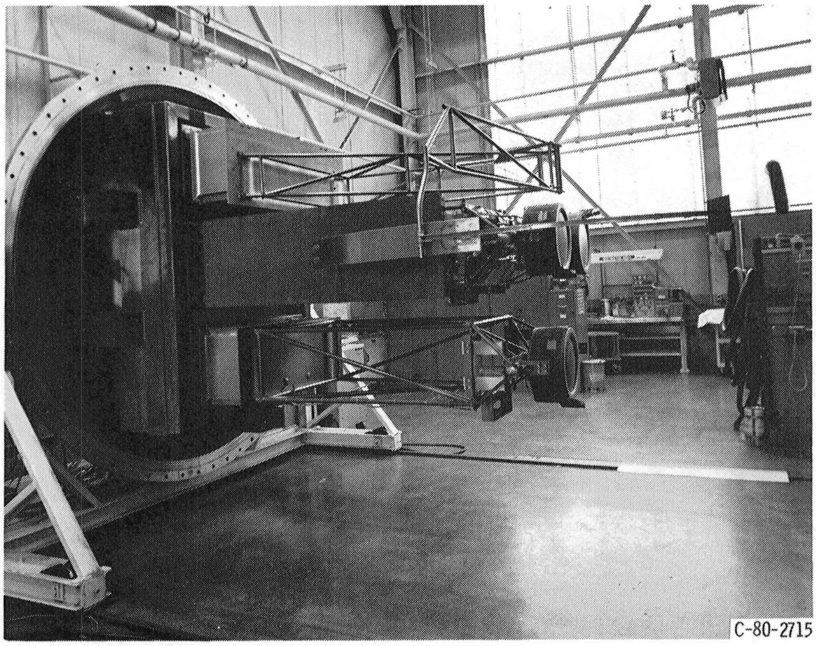


Figure 27. —Bimod test facility.

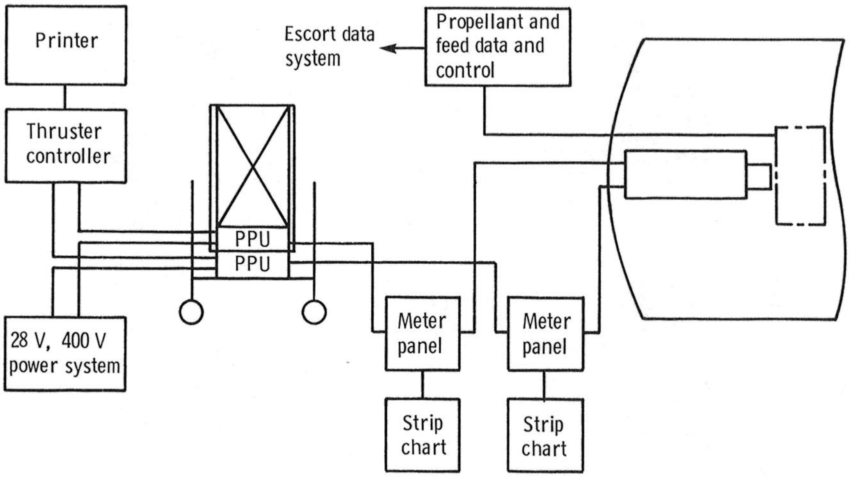


Figure 28. —Bimod tank test.

Lewis. The 30-cm thruster was designed at the Hughes Research Laboratory with strong support from Lewis. The power-processing unit to run the thruster was electrically designed by TRW, but the mechanical and thermal design and fabrication of the five functional model PPU's were done at Lewis. As a part of the technology readiness program, in order to demonstrate realistic hardware, a prototype thruster subsystem was fabricated and tested at Lewis. This subsystem provided realistic weights, thermal characteristics, and electrical performance. This subsystem design included gimbals, a propellant supply and distribution system, and thermal radiators.

This program developed thruster manufacturing expertise at Hughes for the design, fabrication, testing, and documentation of the J-series thruster. TRW designed the circuits and components for the PPU and fabricated an electrical breadboard PPU. In addition they supplied all the electrical components from which Lewis fabricated the functional-model PPU's.

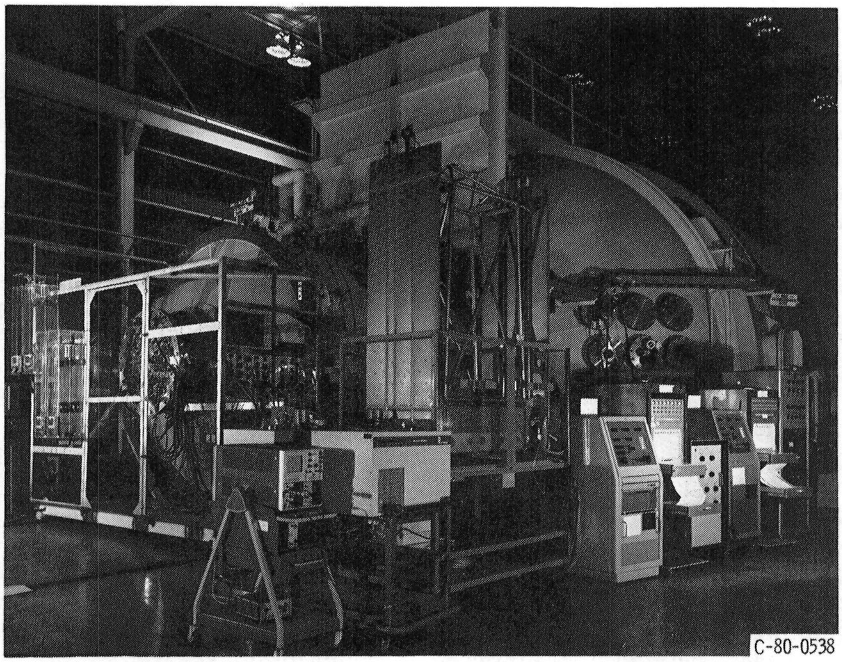


Figure 29. – Bimod test facility instrumentation.

Technology transfer mechanisms. – The major mechanisms used to transfer the electric propulsion technology from the Lewis Research Center to the users include

(1) Design manuals
(2) Industry briefings
(3) Technical society presentations
(4) Direct contractor involvement
(5) Support of MSFC phase B contractor efforts
(6) Participation and interaction at contractor reviews
(7) Weekly teleconferences
(8) Training classes

Technology transfer chronology. – In April 1979, Lewis and MSFC began activities to transfer the 30-cm ion thruster technology in the most expeditious manner at the earliest time consistent with mission success. The first step in implementing the technology transfer was the preparation of the 30-Centimeter Ion Thrust Subsystem Design Manual (ref. 16), which was published in June 1979. This report documents the designs of the baseline technology program elements and makes available many details previously unpublished. A confidential microfiche supplement contains 7000 pages of text and 600 microfilm drawings.

The design manual was released at an industry briefing held at Lewis on June 14 and 15, 1979. About 100 people from 25 contractors and Government agencies attended.

In July 1979, a memorandum of understanding was established between MSFC and Lewis to support the SEPS phase B contract development. In this memorandum Lewis's responsibilities were stated as:

(1) To participate in the SEPS phase B source evaluation process
(2) To participate in the negotiations of the study plans to be implemented by the selected phase B contractors
(3) To participate in contractors' reviews and to provide written comments
(4) To provide status reviews of the technology readiness program in January 1980 and the fourth quarter of 1980
(5) To familiarize the MSFC technical team and the phase B contractors with the operation of the ion propulsion system as part of the technology transfer process

Lewis was also to assign personnel for execution of the tasks defined in the memorandum and to identify a point of contact for technical information request.

To support the SEPS phase B source evaluation process, Lewis provided one member of the Source Evaluation Board (SEB). The Lewis Technical Consultant to the Director of Space Systems and Technology was assigned to the SEB. He also later supported the negotiation of the phase B contractor study plans. The head of the Lewis Technology Section was assigned as a member of the SEB Technical Evaluation Committee.

To expedite transfer of the 30-cm ion thruster technology, weekly teleconferences were held between MSFC and Lewis to keep MSFC totally informed on all facets of the 30-cm technology readiness program. Also, several meetings were held at Lewis with MSFC to further clarify technical information contained in the design manual.

MSFC awarded two phase B study contracts in December 1979, and Lewis supported the initial kickoff meetings held at MSFC in January for each phase B contractor. Following the meeting at MSFC each phase B contractor attended a meeting at Lewis to describe his understanding of the PPU-thruster requirements. At this time Lewis personnel clarified further requirements and provided design guidelines based on their expertise.

On January 31, 1980, a technology readiness status review was held at Lewis to update the SEPS phase B participants on the progress of the technology readiness program since the initial briefing in June 1979. Splinter meetings were held with each phase B contractor the following day. As a result of these splinter meetings, Lewis developed a thruster requirements document to ensure understanding by all participants.

In March 1980, Lewis supported the first phase B contractor review at MSFC and provided strong participation in the splinter sessions.

During the week of April 15th, 1980, the familiarization-training course called out in the MSFC-Lewis memorandum of understanding was held at Lewis. The seminar included lectures and "hands on" hardware experience with two different thruster test setups.

Both SEPS phase B contractors supported independent research and development programs to demonstrate their proposed ion thruster power-processing concepts. In addition, Lewis made contractual arrangments within the technology program to support these efforts. One phase B contractor was loaned a 30-cm improved PPU being developed by Lewis. The other phase B contractor was provided access to an ion thruster test facility. Both contractors were provided with 30-cm ion thrusters to use in performing integration tests with their power processors.

In June 1980, Lewis supported the SEPS midterm review at MSFC in accordance with the memorandum.

A reassessment of the technology transfer process was made in July 1980. MSFC was making major strides in understanding the 30-cm technology and was participating in the technology readiness program. To further expedite this transfer, a plan was developed to transfer the outstanding contracts in the technology readiness program to MSFC. Provisions were also made for direct participation by MSFC in the in-house systems particle and fields testing.

Final documentation and a second industrial briefing on the 30-cm technology program completed the technology transfer program.

Summary

Electric propulsion has made enormous technological strides since its inception in the late 1950's.

Research and technology efforts at Lewis have focused on one concept, the mercury-bombardment ion thruster, taking it from a laboratory experiment to realistic flight hardware. Flight tests demonstrated thrust and neutralization (SERT I) and long-term operation, orbit raising, and cross-neutralization (SERT II). Laboratory life tests demonstrated thruster life of 15 000 hours and component life in excess of 30 000 hours. Multiple thruster systems demonstrated the interface and operation of complete systems.

Performance and reliability have been demonstrated to levels adequate to enable missions that are impossible to perform by any chemical propulsion techniques. Mission opportunities like comet and asteroid rendezvous, which were heretofore prohibitively expensive to attempt by conventional techniques, are now within our grasp.

The technology to perform these very difficult and advanced missions is radically different from the technology of chemical systems. To transfer this technology required a program to familiarize the potential user with the characteristics and applications of electric propulsion hardware.

The Lewis Research Center, which had developed electric propulsion technology since its inception, was chosen to bring the technology for an ion thruster system to a state of technology readiness and to devise a process to transfer this technology to a user Center, the Marshall Space Flight Center (MSFC). Building on the foundation of the defunct Advanced Systems Technology (AST) program, Lewis defined the technology readiness criteria and established a program to accomplish the task. Close technical and managerial relationships were established with the MSFC to indoctrinate their technical people with the results of the technology readiness program. In addition, Lewis published a design manual detailing all the critical aspects of the technology and presented these data to NASA and industry by means of a series of design briefings.

The Lewis program was completed in 1980. The technology was demonstrated and MSFC took over the task of developing the flight system. The only step remaining is the approval of a space flight

project to use this technology. At this writing, the prospects for a near-term project approval are not favorable; but, when a project is approved, the technology and the developers will be ready.

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16. Abstract

This report describes a sequence of events, occurring over the last 25 years, that chronicle the evolution of ion-bombardment electric propulsion technology. Emphasis is placed on the latter phases of this evolution, where special efforts were made to pave the way toward the use of this technology in operational space flight systems. These efforts consisted of a planned program to focus the technology toward its end applications and an organized process that was followed to transfer the technology from the research-technology NASA Center to the user-development NASA Center and its industry team. Major milestones in this evolution, which are described, include the development of thruster technology across a large size range, the successful completion of two space electric rocket tests, SERT I and SERT II, development of power-processing technology for electric propulsion, completion of a program to make the technology ready for flight system development, and finally the technology transfer events.

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